# NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

## TECHNICAL NOTE 2131

## BOUNDARY-LAYER TRANSITION ON A COOLED 20° CONE AT MACH NUMBERS OF 1.5 AND 2.0

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## SUMMARY

The laminar boundary layer on a cooled 20° cone was investigated at Mach numbers of 1.5 and 2.0 to determine the variation of the position of transition with surface temperature for both constant surface temperatures and surface temperatures that increased toward the rear of the cone. Small circumferential grooves at the tip were necessary to cause transition near the center of the cone at the condition of zero heat transfer. The results obtained at all test conditions indicate that removing heat from the boundary layer delays transition from laminar to turbulent flow. The theory of laminar-boundary-layer stability developed by Lees in NACA TN 1360, 1947, is in qualitative agreement with this result. At the test conditions of the present investigation, the increase in the extent of the laminar boundary layer was found to be a linear function of the rate of heat transfer.

## INTRODUCTION

The theoretical work by Lees, reported in reference 1, indicates that heat transfer has a marked effect on the stability of laminar boundary layers at supersonic speeds. The theoretical effect of adding heat from a surface to a laminar boundary layer is to decrease its stability; whereas extracting heat from a laminar boundary layer tends to increase its stability. This latter effect is of considerable importance in the design of supersonic aircraft because of the possibility of obtaining low friction drag coefficients and reduced aerodynamic heating by extending the laminar-flow region in the boundary layer.

The experimental results reported in reference 2 indicate that adding heat to laminar boundary layers causes premature transition. The case of heat removal, however, is believed to be of more practical importance than that of heat addition, and additional experiments are necessary to determine to what extent removing heat will stabilize the boundary layer.

An exact experimental comparison with Lees' theory would require measurements of the amplification or damping of known disturbances in a laminar boundary layer similar to the measurements made at subsonic speeds by Schubauer and Skramstad (reference 3). Results that are useful for most engineering purposes, but not directly comparable with the theory, can be obtained by measuring the effect on transition caused by removing heat from the boundary layer. The present experiments were undertaken to investigate this effect.

Since in most applications the surface temperature cannot be expected to be uniform, it was decided that both the case of uniform surface temperatures and the case of a positive surface-temperature gradient (surface temperature increasing toward the rear of the cone) would be investigated.

## NOTATION

$H_o$ total pressure in tunnel settling chamber, pounds per square inch absolute

$L$ body length, feet

$M$ Mach number, dimensionless

$r$ recovery factor $\left(r = \frac{T_r - T_v}{T_o - T_v}\right)$, dimensionless

$T_r$ recovery temperature (surface temperature $T_s$ at condition of zero heat transfer), $^{\circ}F$

$T_o$ total temperature, $^{\circ}F$

$T_s$ surface temperature, $^{\circ}F$

$T_{s_\infty}$ theoretical surface temperature for infinite boundary-layer stability, $^{\circ}F$

$T_v$ static temperature just outside the boundary layer, $^{\circ}F$

$x$ distance from nose along body axis, feet

## APPARATUS

The experimental investigation was made with an air-cooled 20° cone in the Ames 1- by 3-foot supersonic wind tunnel No. 1. This closed-circuit variable-density wind tunnel is equipped with a nozzle having flexible top and bottom plates which can be shaped to give test-section Mach numbers in the range of 1.2 to 2.4. The total-pressure level in the wind tunnel can be varied from one-fifth of an atmosphere to three atmospheres absolute, depending upon the Mach number and the ambient-air temperature. The air in the wind tunnel is dried to an absolute humidity of 0.0001 pound of water per pound of dry air in order to make the effects of condensation in the nozzle negligible.

The cone, which is shown in figure 1, was installed on a sting in the center of the test section, and the coolant was brought to and from the cone through a thin cantilevered strut mounted behind the cone on the side wall of the wind tunnel.

### Air-Cooled Cone

The test body consisted of two concentric cones with a narrow annular passage between them. Both the cones were made of stainless steel and the outer cone had a wall thickness of only 0.028 inch in order to minimize heat conduction along the surface. The surface of the outer cone was highly polished in order to prevent localized transition due to roughness and to reduce radiant heat transfer. The radiant heat transfer of an almost identical body at similar test conditions has been found to be negligible. (See reference 2.)

Cooling air entered through the tube connected to the hollow sting and flowed forward through the center tube to the cone tip. From the chamber at the tip, the air flowed aft through the narrow annular passage between the inner and outer cones into the rear chamber and out the exhaust tube. Two inner cones were employed in the investigation: one was designed to give an air passage such that a constant surface temperature would be obtained in the test region $(x/L = 0.3 \text{ to } 0.8)$, and the other was designed to give surface temperatures that increased toward the rear of the cone.

Another body, with external dimensions identical to those of the cooled cone, was equipped with pressure orifices. This cone was used to obtain measurements of the local pressures which would exist on the surface of the cooled cone during the tests.

### Cooling System

The details of the cooling system are given in reference 4. Dry air from the wind-tunnel make-up air system was used as the coolant, and dry ice in an alcohol bath was used to decrease the coolant temperature. This system provided an adjustable and stable outlet temperature at a constant coolant flow rate, and coolant temperatures down to $-90^{\circ}$ F were obtainable.

## INSTRUMENTATION

Boundary-layer transition at supersonic speeds can be detected by measurements of the longitudinal surface-temperature distribution. The increase in the local heat-transfer coefficients in the turbulent-boundary-layer region over those in the preceding laminar region causes a corresponding change in surface temperature when the longitudinal heat-transfer distribution is reasonably uniform. This method of detecting transition is subject to some inaccuracy due to conduction in the shell of the cone. If the model shell is sufficiently thin, however, the position of transition can be measured with good accuracy.

The surface temperature was measured at 24 points, 12 on each of 2 rays of the cone 180° apart, by constantan-stainless-steel thermocouples. The thermocouples were made by soldering fine constantan wires into small holes drilled through the outer shell. The wires, between the soldered junctions and the cone base, were flattened to a thickness of 0.002 inch and cemented to the inner surface of the outer cone with a thin film of insulating cement. This method of installation minimized the interference of the leads with the flow of air in the coolant passage. As shown in figure 1, the flattened wires were arranged at the base so that a sliding contact was made with the constantan lead wires. A single stainless-steel wire was used as a common lead and, with the constantan leads, was connected through an automatic switch to a potentiometer which automatically recorded the surface-temperature distribution.

A standard recording potentiometer was modified to allow its use with the constantan-stainless-steel thermocouples, and the measurements of surface temperature with the modified instrument were accurate to ±0.5° F. A second recording potentiometer was used to measure the total-temperature distribution in the wind-tunnel settling chamber to indicate when the wind tunnel was at thermal equilibrium.

## TEST PROCEDURE

In order to measure the effects of cooling, it was necessary to have the transition point near the center of the instrumented area on the cone. However, it was known from previous heat-transfer tests with 20° cones that transition would occur downstream from the instrumented area on a smooth cone at the condition of zero heat transfer at the maximum available Reynolds number. The required forward movement of the transition point was accomplished by the use of three small grooves around the cone at approximately the 8-, 10-, and 12-percent-length stations. All the grooves were 0.010 inch wide and 0.015 inch deep, and extreme care was taken to make them of uniform depth and with uniform sharp edges in an attempt to obtain the same longitudinal position of transition on all rays of the cone.[1]

The first phase of the investigation consisted of measuring the local pressures on the pressure-distribution cone. Then the cooled cone was installed in exactly the same position in the wind tunnel for the transition tests.

The procedure for the transition measurements consisted of first operating the wind tunnel with the cone at the zero-heat-transfer condition ($T_s = T_r$) until the total temperature of the air stream and the cone surface temperature reached equilibrium. Then the surface temperature on the cone, the total-temperature distribution, humidity, total pressure, and static pressure in the test section were recorded. These data were obtained at several surface temperatures below the recovery temperature in successive decrements of approximately $10^\circ$ F. The minimum surface temperatures investigated were the first temperatures at which transition had moved out of the instrumented area of the cone.

The tests were conducted at a wind-tunnel total pressure of 24 pounds per square inch absolute at a Mach number of 1.5, and at 28 pounds per square inch absolute at a Mach number of 2.0. These pressures were selected so that, at the two Mach numbers, the nominal Reynolds numbers based on cone length would be the same (3.75 million).

## RESULTS AND DISCUSSION

The pressure-distribution measurements indicated a small negative pressure gradient at a Mach number of 2.0 and a small irregular pressure variation at a Mach number of 1.5. The maximum pressure variations were less than 1 percent of the free-stream dynamic pressure at both Mach numbers, and are believed to have been constant during tests at each Mach number. Therefore the static pressure variations should have little effect on the test results.

[1]During the preliminary tests required to determine the type and size of roughness required to cause the desired forward movement of transition, it was found that a ring of 0.001-inch-diameter wire located 0.8 inch from the cone tip caused transition at the wire. A similar result was obtained with a sharp, circumferential ridge 0.002 inch high at the same position on the cone. These results, as compared with those obtained with narrow, sharp-edged grooves, indicate that projections above surfaces are the more effective of the two methods of causing a forward movement of transition. For the present investigation, circumferential projections were actually too effective, and a much better control of transition was obtained by the use of narrow grooves.

The results of the surface-temperature measurements are shown in figure 2. Only the data from one side of the cone are presented because transition occurred in a similar manner on the opposite side, but at a position approximately 5 percent of the cone length aft of that shown, at all test conditions. It should be noted that the upper curves were obtained at the condition of zero heat transfer ($T_s = T_r$). The rise in temperature which occurs at transition serves to indicate the movement of transition with decreasing nominal surface temperature. The loci of the points indicating the probable beginning of transition are shown in figure 2.

The fact that the rise in surface temperature is associated with transition has been verified in four ways:

1. Liquid-film tests at recovery temperature have indicated transition in the same region as the rise in surface temperature.

2. The surface-temperature distributions ahead of the rise in surface temperature are almost identical with those shown in reference 4 for the same model with a laminar boundary layer.

3. The region in which the rise in surface temperature occurs moved forward with an increase in the number of grooves.

4. The recovery factor increased by 0.04 in the region where the rise in surface temperature occurs[2], which is in agreement with both theory and other experiments. (See reference 5.)

The variation of the length of laminar run along the cone with surface temperature is approximately linear for both Mach numbers and both surface-temperature gradients and indicates increasing boundary-layer stability with decreasing surface temperature. The theory of reference 1 is in qualitative agreement with this result. At the present test conditions, the theory indicates that the boundary layer should be infinitely stable at surface temperatures of $-6.6^{\circ}$ F and $49.0^{\circ}$ F at Mach numbers of 1.5 and 2.0, respectively. (See figs. 2(a) and 2(c).) The data shown in figure 2(c) indicate that transition occurred at surface temperatures below the theoretical infinite-stability limit at a Mach number of 2.0, and the data shown in figure 2(a) indicate that a similar result is possible at a Mach number of 1.5. This might be expected because the theory, as Lees has pointed out, is not strictly applicable to the prediction of transition.

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[2]The recovery factor on this model, without grooves, was 0.85, which is in agreement with the results for a laminar boundary layer reported in reference 5. The absolute values of the recovery factors ahead of and behind the rise in surface temperature were found to decrease continuously with an increase in number of grooves but their difference was always $0.04 \pm 0.005$. With the three grooves that were used, the laminar-boundary-layer recovery factor was 0.81 and the turbulent-boundary-layer value was 0.85.

The limited range of the experiments makes it impossible to draw any definite conclusions regarding the effects of Mach number on the effectiveness of cooling as a means of delaying transition. The data indicate, however, that the rates of movement of transition with surface temperature are at least similar at both Mach numbers. The constant slopes of the lines through the points at the beginning of transition for the case of uniform surface temperatures (figs. 2(a) and 2(c)) indicate that, for the present experiments in which roughness was used near the cone tip, the increase in the length of laminar run is directly proportional to the difference between the surface and recovery temperatures, and therefore is a linear function of the rate of heat transfer.

The slopes of the lines indicating the loci of the beginning of transition for the case of positive surface-temperature gradients are very nearly equal to the slopes for the case of uniform surface temperatures. It should be noted, however, that because of the appreciably greater differences between the surface and recovery temperatures ahead of transition for the case of positive surface-temperature gradients, the heat to be removed to delay transition by a given amount is greater than for the case of uniform surface temperatures.

## CONCLUDING REMARKS

The effect of removing heat from the boundary layer is to delay transition. The increase in the extent of the laminar boundary layer for uniform surface temperatures was found to be directly proportional to the difference between the surface and recovery temperatures, and therefore a linear function of the rate of heat transfer, at the test conditions of the present investigation. A somewhat greater rate of heat removal was required to delay transition by a given amount for the case of positive surface-temperature gradients than for the case of uniform surface temperatures.

Ames Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Moffett Field, Calif., Apr. 17, 1950.

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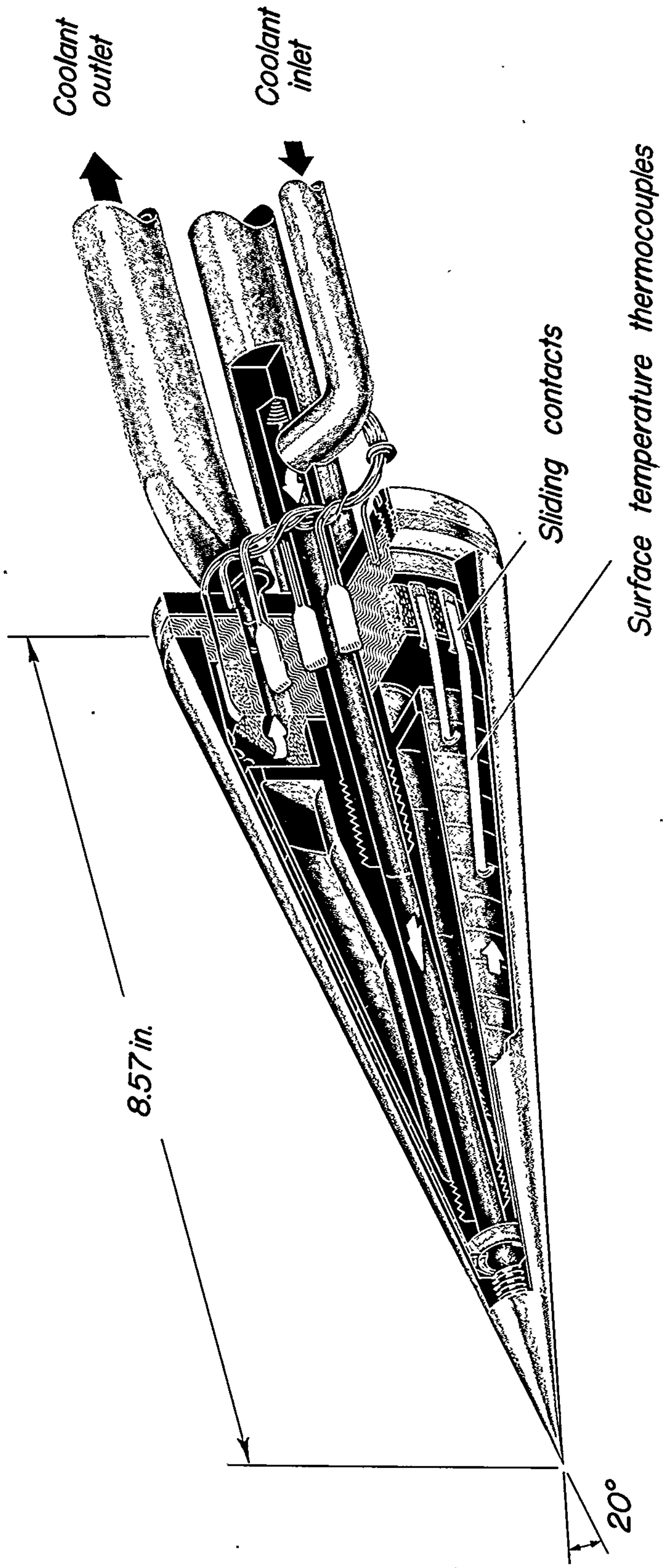
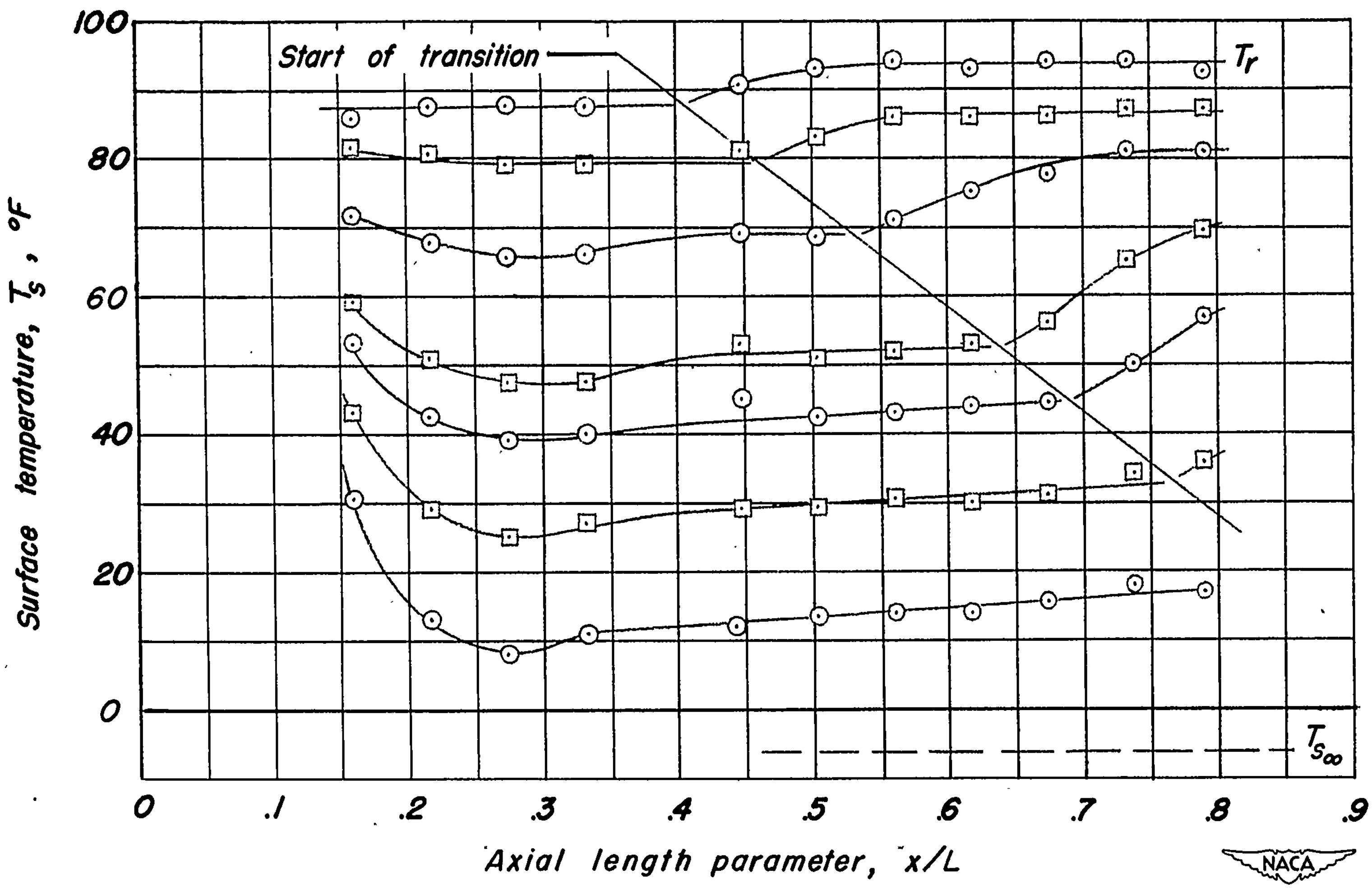
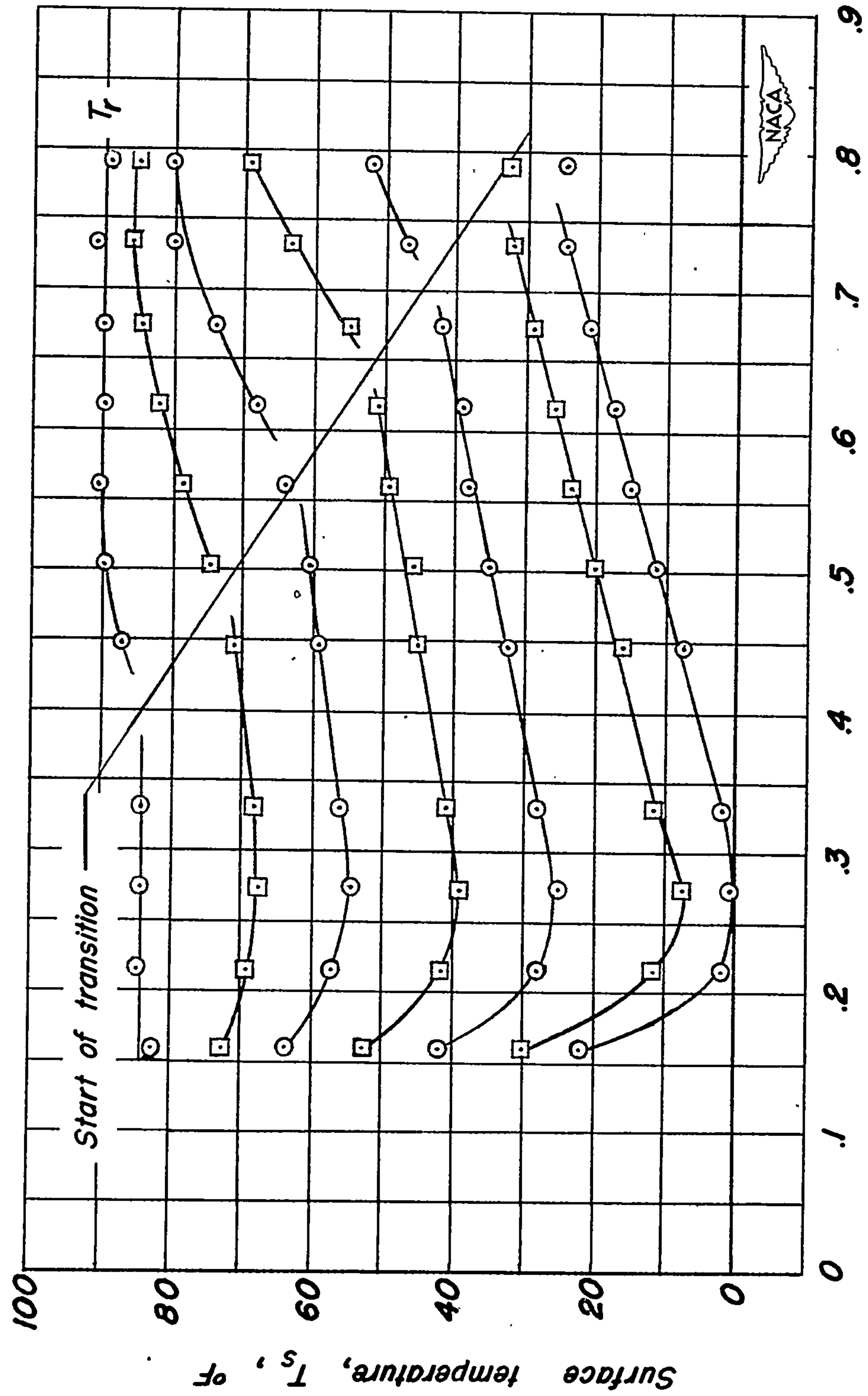



Figure 1. Air-cooled cone.



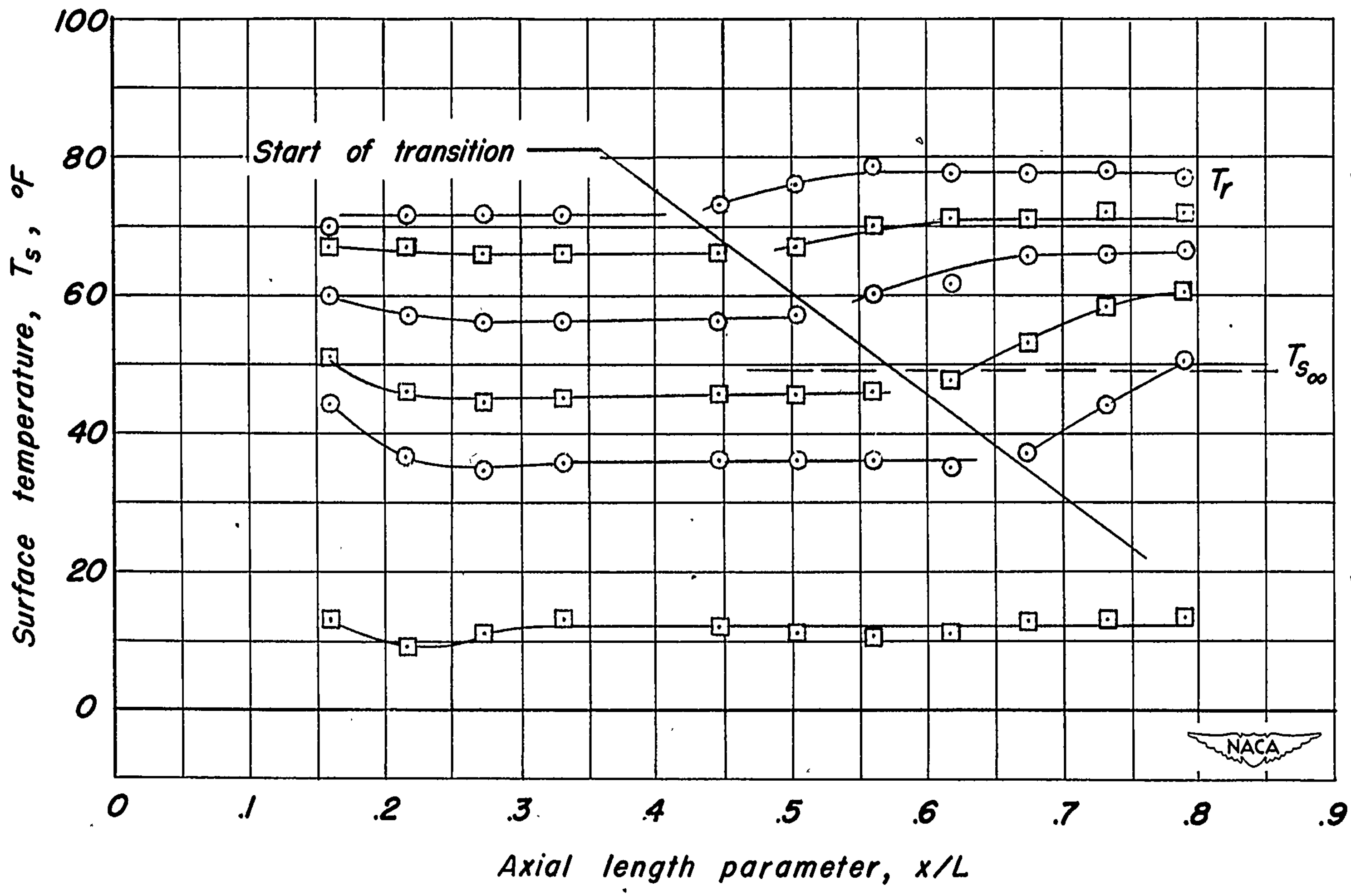

(a) Zero temperature gradient, $M = 1.5$, $H_0 = 24$ psia.

Figure 2.—Surface temperature distributions for various nominal surface temperatures on the cooled 20° cone.



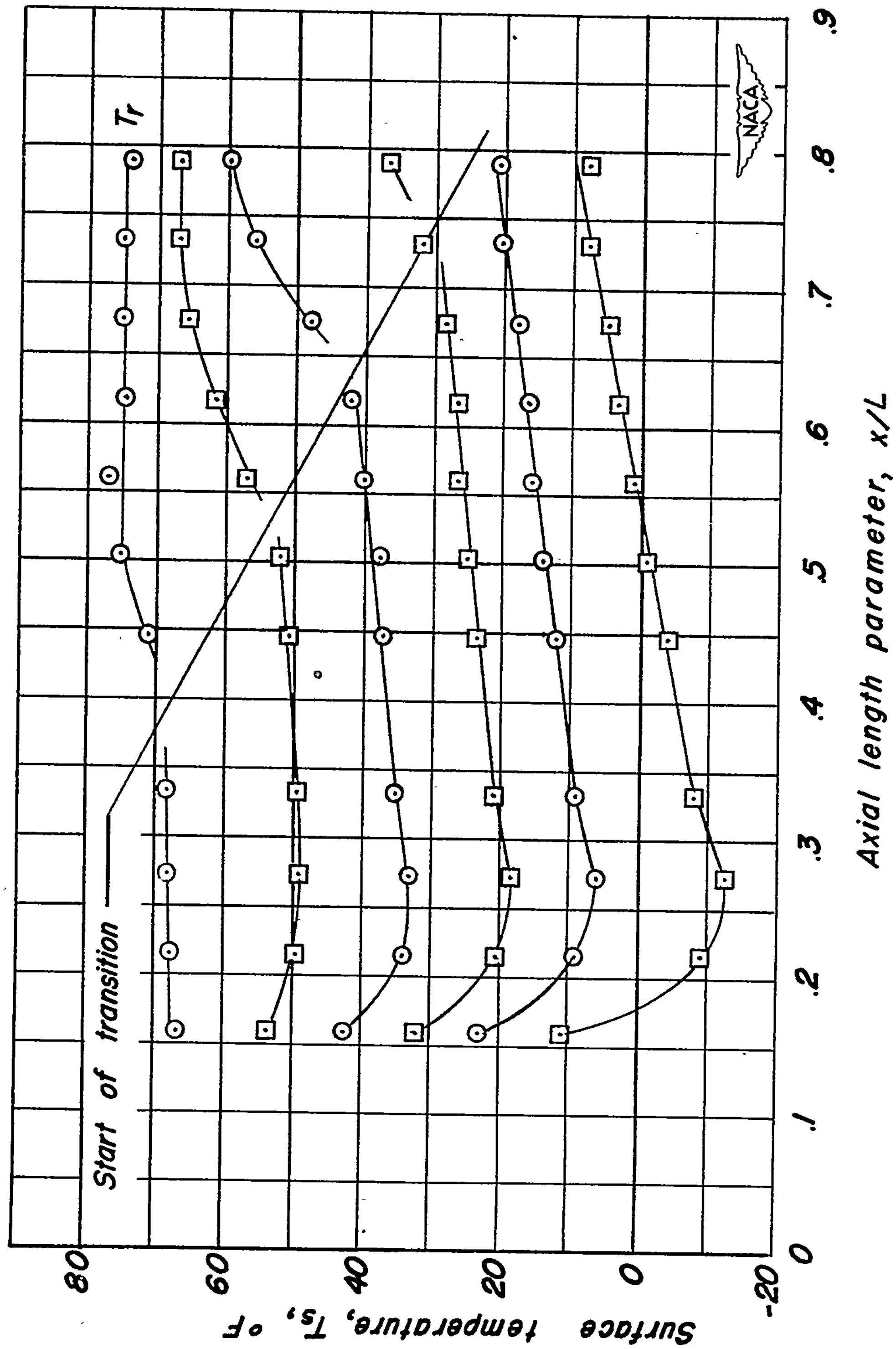

(b) Positive temperature gradient, $M=1.5$, $H_0=24$ psia.

Figure 2. — Continued.




(c) Zero temperature gradient, $M = 2.0$, $H_o = 28$ psia.

Figure 2. — Continued.




(d) Positive temperature gradient, $M = 2.0$, $H_o = 28$ psia.

Figure 2.—Concluded.